Advanced Electric Propulsion for RLV Launched Geosynchronous Spacecraft

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RLV LAUNCHED GEOSYNCHRONOUS SPACECRAFT

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ABSTRACT

Solar Electric Propulsion (SEP) when used for station keeping and final orbit insertion has been shown to increase a geostationary satellite's payload when launched by existing expendable launch vehicles. In the case of reusable launch vehicles or expendable launch vehicles where an upper stage is an expensive option, this methodology can be modified by using the existing on-board apogee chemical system to perform a perigee burn and then letting the electric propulsion system complete the transfer to geostationary orbit. The elimination of upper stages using on-board chemical and electric propulsion systems was thus examined for GEO spacecraft. Launch vehicle step-down from an Atlas IIAR to a Delta 7920 (no upper stage) was achieved using expanded on-board chemical tanks, 40 kW payload power for electric propulsion, and a 60 day elliptical to GEO SEP orbit insertion. Optimal combined chemical and electric trajectories were found using SEPSPOT. While Hall and ion thrusters provided launch vehicle step-down and even more payload for longer insertion times, NH3 arcjets had insufficient performance to allow launch vehicle step-down. Degradation levels were only 5 to 7 percent for launch step-down cases using advanced solar arrays. Results were parameterized to allow comparisons for future reusable launch vehicles. Results showed that for an 8 W/kg initial power/launch mass power density spacecraft, 50 to 100 percent more payload can be launched using this method.

INTRODUCTION

Solar Electric Propulsion (SEP) is currently being used for station keeping of geosynchronous satellites. Examples include hydrazine arcjets on Lockheed Martin spacecraft, ion thrusters on Hughes spacecraft and Hall thrusters on Russian spacecraft.[1] Combined with this use of SEP is the continuing trend for geosynchronous spacecraft towards longer lifetimes, increased masses, higher powers, and increased service bandwidth. The next step is to combine this growth in spacecraft power with SEP to assist with the delivery of the spacecraft to geosynchronous orbit (GEO) by using high earth starting orbits. This concept has been shown to be advantageous in terms of net mass by several authors and is currently being offered to satellite buyers to increase payload mass.[2–9] In this context, net mass refers to the total spacecraft mass minus the wet propulsion system mass and any power system mass added only for propulsion. In most previous studies the SEP starting orbits were not optimized.

Previous works showed the benefits of advanced SEP technology using optimized SEP starting orbits for the various expendable, upper staged launch vehicles with planned, high powered (10-25 kW) spacecraft.[7,8] The purpose of this paper is to expand this work to enable a launch vehicle step-down, specifically moving from an Atlas IIAR class launch vehicle to a Delta II 7920 class vehicle or reusable launch vehicle, neither of the latter having an upper stage. This paper describes the mission analyses, propulsion options and optimized trajectory results to achieve the launch vehicle step-down. The SEP system also performs fifteen years of station keeping. In the previous two works optimal trajectories were found for 10, 15, 20, and 25 kW powered spacecraft. A payload power level of 40 kW was assumed available for the electric propulsion orbit transfer in this work. This power is representative for the next generation of geosynchronous communications satellites.

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In the previous studies the mass impact of replacing some portion of the chemical apogee propulsion system with a SEP system was considered. Those expendable launch vehicles had an upper stage to lift the payload at least to geostationary transfer orbit if not geostationary orbit. Fuel was then off-loaded from the on-board chemical system or upper stage to start at a lower orbit and allow the electric propulsion system to finish the transfer to geostationary orbit. In the case of reusable launch vehicles or some expendable launch vehicles where an upper stage is an additional, expensive option, this methodology can modified by using the existing on-board apogee chemical system to perform a perigee burn and then letting the electric propulsion system complete the transfer to geostationary orbit. Use of this method will allow the nominal payloads of larger expendable launch vehicles to be moved to smaller and cheaper expendable launch vehicles or reusable launch vehicles without expensive chemical upper stages. In this work, three electric propulsion technologies are considered: NH3 arcjets, xenon Hall, and xenon Ion thrusters.

MISSION ANALYSIS, OPTIONS AND ASSUMPTIONS

Mission Analysis

The approach is to utilize the numerical optimizer SEPSPOT with its option to perform optimal impulsive stage analysis to minimize the SEP transfer time.[10] All that is required for the high thrust portion of the program is a final mass for this portion of the mission and an initial impulsive ΔV. The final mass of the impulsive portion is the starting mass for the SEP mission. The ΔV is the velocity or energy change required for an orbit transfer. Impulsive ΔV assumes an instantaneous burn and is assumed for all the chemical propulsion burns in these analyses. The SEP transfer mission ΔVs differ from impulsive due to gravity losses associated with constant thrusting and nontangential steering.[11]

The expendable launch vehicle assumed for this analysis is the Delta 7920 which does not use a third stage.[12] The vehicle was selected as representative of mass delivery, but was not assessed with respect to other integration issues. The Delta 7920 places the payload satellite, including the necessary on-board propulsion systems to achieve geostationary orbit, into a 185 km altitude circular parking orbit. The starting mass in the parking orbit is 5089 kg which includes the spacecraft and the on-board electric and chemical propulsion systems. After reaching parking orbit the on-board chemical stage, normally used for apogee insertion, is used to lift the spacecraft to a starting orbit for the electric propulsion. The amount of fuel available to the on-board chemical system is varied to allow different SEP starting orbits.

Two state-of-art (SOA) mission cases in which an electric propulsion system performs only station keeping are used as baselines for comparison purposes. It uses either a Delta 7925 with a solid upper stage or an Atlas IIAR with cryogenic upper stage to place the spacecraft into geostationary transfer orbit (GTO) and the on-board chemical system to insert itself into geostationary orbit.[12] The end-of-life net masses possible with the Delta 7925 and the Atlas IIAR are calculated to be 840 kg and 1660 kg, respectively. The other mission cases use a varied on-board chemical fuel mass, which gives a set ΔV, in an optimal one or two burn transfer to an optimal SEP starting orbit as shown in Figure 1. The on-board chemical portion of this transfer is not necessarily to GTO.

The SEPSPOT program determines the required one or two impulsive burns with the allotted ΔV to reach an SEP starting orbit which minimizes the SEP trip time. This SEP starting orbit can have any perigee, apogee, and inclination combination which is achievable with the given impulsive ΔV. This ΔV is the ΔV capability of the on-board chemical system which is varied in specific cases 1 to 17 (Fig. 2).
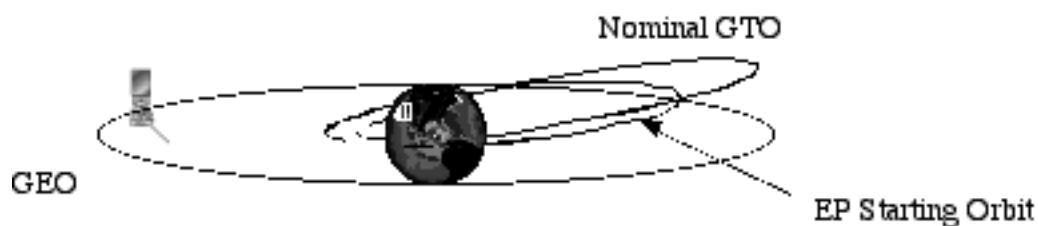

Figure 1.—Sample SEP starting orbit.

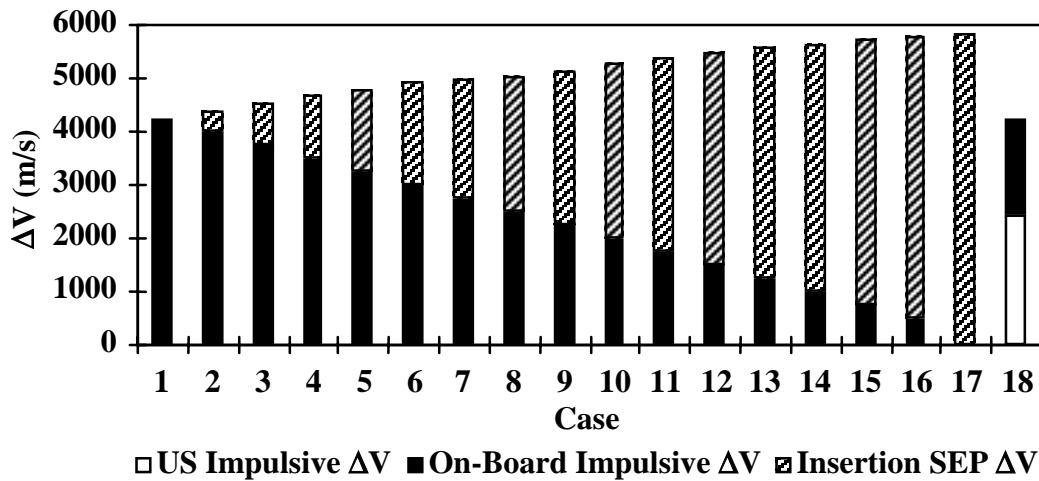## Mission ΔV Breakdowns for 40 kW H-Hall, Cases 1-18

Figure 2.—Orbit insertion mission ΔV breakdown.

This on-board chemical ΔV portion is varied from 4270 to 520 m/s in 250 m/s steps (cases 1 to 16) to show the trade between increased net mass and increased trip time. An all SEP case is also run to represent the 0 m/s ΔV case (case 17). To illustrate these trades, Figure 2 shows a variation between the on-board chemical ΔV and the transfer SEP ΔV for a case using Hall thrusters (results are similar for other thruster options). The required SEP ΔV from SEPSPOT to replace the on-board chemical ΔV is greater due to gravity losses. This required SEP ΔV is further discussed in the results section. The mass of the satellite after all the allotted chemical fuel is used is the starting SEP phase mass. Each SEP technology is traded for each case.

The SEP phase optimization includes the impacts of shading, J2 (Earth oblateness), and the solar array degradation due to Van Allen belt radiation. Unfortunately, SEPSPOT does not account for atmospheric drag, which for low starting orbits would have an impact. The SEP system parameters of initial power level, I_{sp}, and efficiency are fixed in the SEPSPOT program. The SEPSPOT program assumes continuous thrusting except while the spacecraft is in shade. SEPSPOT finds the optimal steering for the minimum time trajectory.

The impact of power degradation on the trip time causes SEPSPOT to minimize time spent in the Van Allen belts. As power is degraded, SEPSPOT throttles the thrusters accordingly while maintaining the same I_{sp} and efficiency. While thruster performance normally varies as a function of power level this effect s neglected in this work. The impacts of non-optimal steering and guidance, navigation, and attitude control limitations are not considered here. The impacts of these issues are typically minor.

In addition to the transfer, fifteen years of north/south station keeping (NSSK) are assumed for all cases.[1] While the yearly ΔV varies with satellite station longitude, 45.37 m/s is chosen as representative.[13] East/west station keeping requirements are an order-of-magnitude smaller than NSSK requirements and are neglected in these analyses.

SYSTEM ASSUMPTIONS AND MODELING

On-Board Chemical Propulsion System
For mission scenarios requiring an on-board chemical propulsion system for some part of the orbit insertion, a SOA 328 s I_{sp} bipropellant system is assumed.[1] The system has a fixed dry mass of 23 kg and a tankage fraction of 0.08. The SOA chemical system is deleted from the spacecraft for those missions where the SEP system performs the whole mission.

On-Board Electric Propulsion System

The following technologies are considered in this work: 1.8 kW N_2H_4 arcjets[14] for the baseline case where SEP is for NSSK function only, 10 kW NH3 arcjets,[15] 10 kW xenon Hall thrusters[16,17] and 10 kW xenon ion thrusters.[18] The power given is the power into the power processing unit (PPU). All thruster parameters are shown in Table 1. The M– and H–cases represent two potential Hall thruster operational setpoints. Except for the case of the 1.8 kW arcjet, used in the SOA NSSK only case, four 10 kW thrusters are placed on the spacecraft for orbit insertion and NSSK. Only the 10 kW Hall thruster is currently in development in several forms.[16,17] The NH3 arcjet is a 10 kW extrapolation based on the 30 kW ESEX arcjet which was successfully flown and tested.[15] The 10 kW ion thruster is also an extrapolation based on the 2.5 kW NSTAR design and performance.[18]

Each thruster unit includes structure, gimbal (except SOA arcjet), and controller. The resulting masses are shown in Table 1. A tankage fraction of 0.07 was used for arcjets and 0.10 for the Hall and ion thrusters. Thruster lifetime is also considered and extra thrusters are added when necessary. PPU lifetime was assumed adequate for both the transfer and station keeping missions.

Fifteen years of north/south spacecraft station keeping is performed by four thrusters, one pair placed on the north face and the other on the south face as shown in Figure 3. These thruster pairs are canted to 17, 45, and 30 degrees for the arcjets, Hall thrusters, and ion thrusters, respectively, from the vertical to minimize plume interaction with the array.[7] The thrusters are gimbaled to the appropriate cant angle for the 10 kW orbit insertion thrusters after orbit insertion is completed. The equivalent NSSK thruster I_{sp} is adjusted for the thruster cant cosine loss are shown in Table 1. To perform the north/south station keeping either the south or north pair is fired about the appropriate orbit node on the order of tens of minutes. If one thruster fails the opposite set are tasked with all NSSK burns.
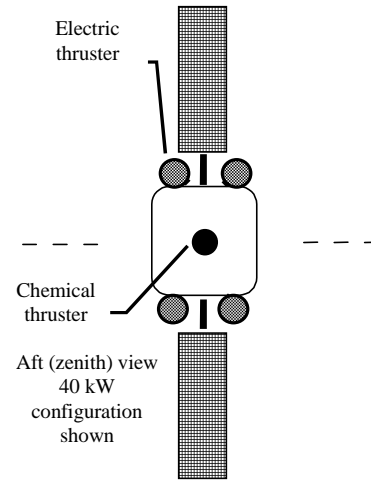

Figure 3.—SEP configuration.

Table 1.—SEP propulsion system parameters.

	SOA N_2H_4 arcjet	NH3 arcjet	Xenon hall thruster M– and H–	Xenon ion thruster
PPU input power, kW	1.8	10	10	10
I_{sp}, s	500	800	1800, 2400	3000
Overall efficiency	0.33	0.27	0.50, 0.55	0.60
Tankage fraction, percent	7	0.07	0.10	0.10
Thruster life, hr	1,000	1,500	7000	8000
Cant angle for NSSK, degree	17	17	45	30
Equivalent cant I_{sp}, s	478	765	1273, 1697	2598
Thrust module thruster, kg	1	3	9	11
Gimbals, percent of thruster	34	25	25	25
Structure, x-percent of gimbals and thrusters	31	31	31	31
Propellant distribution and controller (kg/thruster)	0.55	1.8	1.8	2.8
Total thruster + gimbals + support + propellant dist. (kg/thruster)	2.3	6.7	16.5	20.8
Interface module PPU, cabling kg/kW	2.4	1	3	4
Thermal system (92 percent PPU) kg/kWt-disp	31	13	13	13
Structure, percent of interface components	4	4	4	4
Total PPU + cabling + thermal kg/kWe	6.1	2.1	4.2	5.2

Power System

Advanced solar arrays which provide payload power in geostationary orbit are assumed to provide the 40 kW for the thruster operation during the SEP orbit transfer since the payload is inactive during this phase. This power level was chosen as representative of next generation power levels for geostationary communication satellites (the battery system is assumed to power NSSK thruster operation while the payload uses direct solar array power as suggested by Free).[19] Extra batteries may be required to support the increase in charge/discharge cycling, but this mass is not determined here.

The use of advanced solar cell and array technologies is key to these concepts since the arrays must be light, high power and provide some degradation resistance during the short transit of some part of the radiation belts. Many new array technologies are being developed or are available for use including multijunction cells, thin films arrays, and concentrator arrays.[20] All of these advanced arrays claim improved radiation resistance. An example is the SCARLETT concentrator array which has roughly 44 mils of effective shielding. A trade of solar cell/array technologies is not made here so a representative, high radiation resistance solar array is chosen to have an equivalent layer of 30 mils shielding on the front of the array and infinite shielding on the back of the array for radiation damage calculations.

Since the array is resident on the spacecraft for payload use its mass is not charged to the propulsion system. However, transfer through the Van Allen belts will damage the array. This damaged array mass is charged to the propulsion system at a rate of twice 16.6 kg/kW—once to account for the destroyed portion of the array and a second time to replace the destroyed array for payload use.[21] The replacement array portion could potentially be folded to avoid damage, and deployed on arrival at GEO (An alternative concept would be to add extra array and use it for the propulsion system to allow faster, less damaging transits. This is saved for further studies). Thus, the propulsion system is penalized for long transfers through the Van Allen Belts. The radiation damage that may occur to the payload is not assessed here.

RESULTS

SEP Starting Orbits

Optimal SEP starting orbits determined by SEPSPOT for the 40 kW spacecraft with Hall technology are shown in Figure 4. These SEP starting orbits vary little for the different SEP technologies. The orbit parameters, including apogee altitude, perigee altitude, and inclination, are shown versus the on-board chemical propulsion ΔV. This directly relates to chemical propulsion fuel loading. Only one or two burns are allowed by the code. Cases with 2500 m/s or less of on-board chemical ΔV (cases 8 through 16) use only one perigee burn to lift apogee as high as possible. A slight plane change is also performed. In practice, several perigee burns might be used as well as a small apogee burn to lift the perigee out of notable atmospheric drag. Increasing the on-board chemical ΔV capability above 2500 m/s, allows an optimal two burn case where the apogee is raised above geostationary orbit altitude, the perigee is also raised, and the some portion of the plane change performed. These latter results are similar to those developed earlier.[7,8]
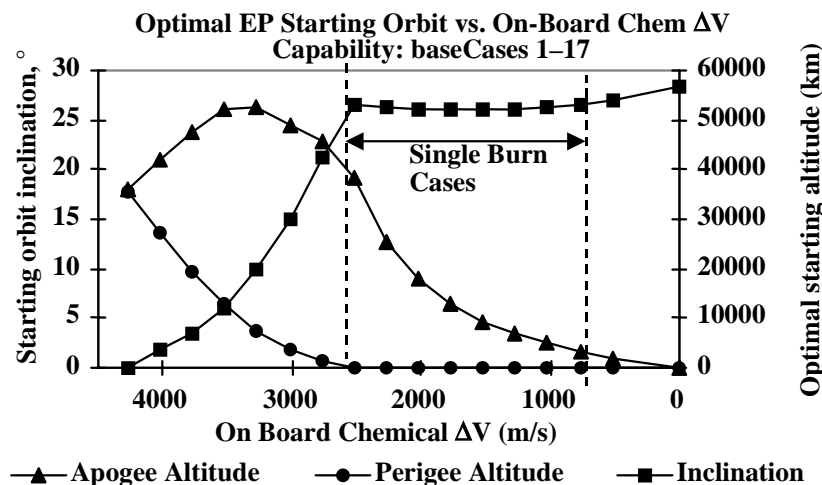

Figure 4.—Optimal SEP starting orbits.

Figure 2 shows the corresponding required transfer SEP ΔV for the varied on-board chemical ΔV for the 40 kW Hall class. These required transfer SEP ΔVs vary little for the different SEP technologies. Cases 1 to 16 show the trade in chemical and SEP ΔV. As on-board chemical ΔV capability is replaced by SEP ΔV, the total ΔV increases due to the losses incurred by the constant thrusting SEP system. Case 17 shows the limit when the LEO to GEO transfer is performed completely by the SEP system. Case 18 represents the baseline LEO to GEO case using a chemical perigee upper stage (US) and an on-board chemical apogee system. While the total mission ΔV is higher using electric propulsion the higher I_{sp} of the SEP system more than offsets this increased ΔV by significantly reducing the total fuel mass. This is shown by the net mass advantage in the next sections.

Figures of Merit
The figures of merit of the advanced SEP systems in this study are the net mass delivered and SEP transfer time. Net mass refers to the usable satellite mass once the wet propulsion system for the orbit insertion and the NSSK missions and any damaged array are removed. It is desired to place the same or greater net mass of a larger launch vehicle onto that of a smaller, cheaper expendable or reusable launch vehicle. In this case the target is the delivery of the baseline Atlas IIAR net mass of 1660 kg onto the Delta 7920.

Launch Vehicle Stepdown
Figures 5 and 6 contain the results of this analysis for a 40 kW class spacecraft for each of the 18 cases in terms of net mass versus SEP transfer time. Each point represents the variation of chemical ΔV as described in the previous sections and is shown in Figure 2. Each case is run with ion, H–Hall, M–Hall, and NH3 arcjets, respectively. Note that the higher the propulsion system I_{sp} the longer each respective fixed chemical ΔV case takes to transfer to GEO. In other words, while all the SEP systems begin at the same starting orbit for each case, the higher thrust systems complete the transfer quicker, though with less payload. The all-SEP LEO to GEO points are shown on Figure 5 as the last point in each data plot (case 17).

A minimum of two months of SEP transfer is needed in order for the electrostatic systems to match the net mass of the Atlas IIAR vehicle as shown in Figure 6. The NH3 arcjet does not have the performance to realize the launch vehicle step-down. Figure 6 also shows that by expanding the electric propulsion system to provide even more of the transfer, even more net mass can be delivered, surpassing that available with the Atlas IIAR. Eliminating the chemical system altogether provides the greatest payload gains but requires trip times over six months.
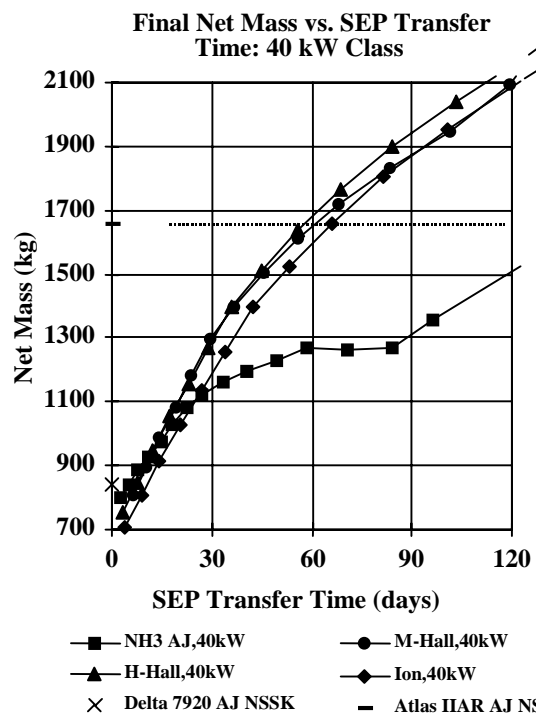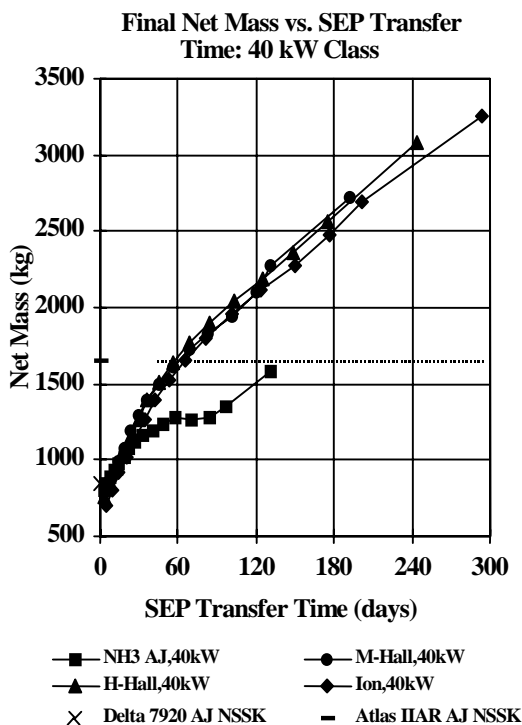
Figures 5 and 6.—Final net mass versus SEP transit time.

For the same on-board chemical ΔV each SEP system requires a different transfer time due to the differences in I_{sp} and efficiency and thus thrust level. The initial steepness of each technology's curve is reduced somewhat for longer transfer times due to the increased rate of solar array damage (Fig. 7), which is subtracted from the net mass.
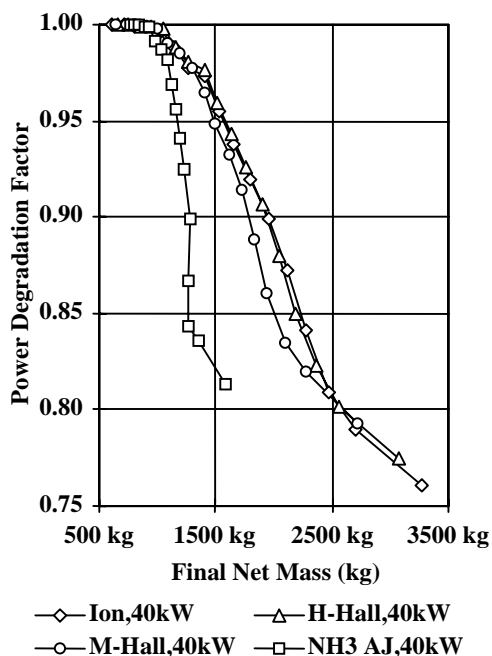

Figure 7.—Power degradation vs. net mass.

This increased damage rate is due starting at lower orbits (higher numbered cases) and thus experiencing longer exposure times in the more damaging portions of the Van Allen belts. For the shortest transfer times, where the on-board chemical system is providing most of the transfer, the radiation damage is small, and the net mass gain increases quickly as allowable SEP transfer time is relaxed. This region of slight degradation occurs for on-board chemical ΔVs above approximately 3000 m/s (cases 1 through 6). Degradation of the solar array for the two month case of interest is only 5 to 7 percent. The degradation versus payload is shown in Figure 7. The accumulated radiation dose on the other spacecraft systems, and its impact is not assessed here. Some radiation hard components and/or shielding may be required.

A sample trajectory for case 10 using the ion system is shown in Figure 8. This case is representative of what is required to achieve the Atlas IIAR stepdown to Delta 7920: 1660 kg of net mass. The ΔV split for case 10 is 2020 m/s on-board chemical and 3270 m/s for the electric thruster. It shows that only a small portion of the inclination is removed by the chemical

system (~1.5 degrees), with the rest being removed at a somewhat continuous rate by the ion system. The chemical system provides only one burn to place the apogee as far above the belts as possible, probably to minimize degradation and/or to reduce plane change ΔV. Thus the transfer orbit remains elliptical, raising both perigee and apogee continuously until finally achieving GEO orbit. Spacecraft power level is also shown in Figure 8. It is clear that the major portion of degradation occurs with the perigee below 10,000 km, or the most damaging part of the radiation belts.
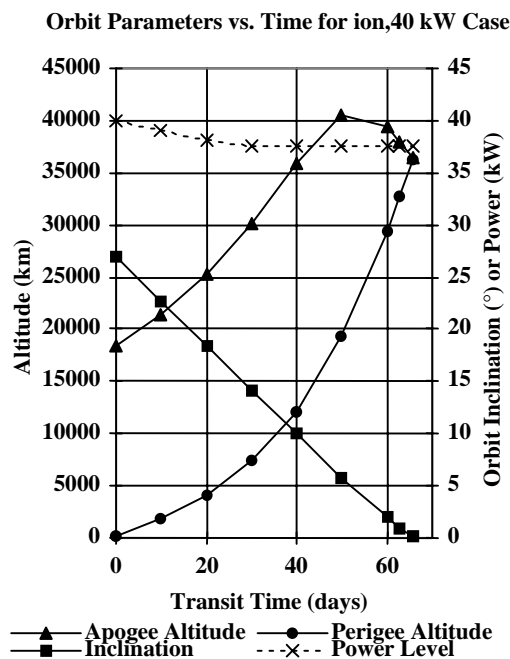

Figure 8.—Orbit parameters for case 10, ion transit from LEO to GEO.

The medium I_{sp} and high I_{sp} Hall systems edge out the ion due to having more optimal I_{sp}/performance for the orbit insertion and being lighter. Work is underway to have a throttleable I_{sp} for future Hall thrusters, so an optimal lower I_{sp} may be used for the orbit insertion and a higher I_{sp} may be used for the NSSK Some additional advantages of the Hall system are smaller size and reduced complexity. The ion system does have the advantage, however, of slightly less radiation exposure for the same delivered payload.

The reality of placing an Atlas class spacecraft on a Delta has some challenges. While the nominal Atlas class net mass can be launched on the Delta 7920 with only a two month transfer time, the on-board chemical system fuel loading would have to be increased over 40 percent. Such stretching of tanks has been done in the past. Without increasing the fuel

loading on an Atlas IIAR class payload a three to four month SEP transfer would be needed, depending on Hall or ion SEP technology chosen, but would deliver a spacecraft with more net mass than the current Atlas IIAR can deliver. Just fitting the larger sized spacecraft on the Delta would require some design modifications. These are not addressed here.

RLV Options

Besides the space shuttle there are many planned one and two stage reusable launch vehicle concepts. Some of these include the NASA/Lockheed Martin VentureStar, the Kistler Aerospace K1, the Rotary Rocket Company Roton, and the Kelly Space and Technology Astroliner with planned launch masses to LEO of 23,000, 5000, 3175, and 5000 kg, respectively.[20] All the reusable concepts share the lack of a built-in upper stage. While upper stages can be added this would add a 'non-reusable' part to the reusability of the system. By using the concept set forth in this work an RLV could place a GEO payload into LEO and return to earth while the payload takes itself to GEO—no extra stages needed (expendable or reusable). Another advantage is the use of the existing payload spacecraft systems (e.g., communications, attitude determination and control, power) to move it from LEO to GEO, much as the GEO spacecraft of today use their own spacecraft systems to take themselves from GTO to GEO.

The results of the past example can be used as a first order estimate on the potential performance for the above mentioned RLVs as well as future concepts. Both the Kistler K1 and the Kelly Space and Technology Astroliner have payloads close to that of the Delta 7920 so the previous example would apply to them. For other RLVs a parameterization of the results given earlier can show what type of power densities would enable the concept. In this case an initial power/LEO starting mass (Pi/Mleo) is defined. The LEO starting mass is equivalent to the spacecraft mass along with all the wet propulsion system mass to take it from LEO to GEO. For the 40 kW example the Pi/Mleo is roughly 8 W/kg. Results are shown in Figure 9.

Figure 9 shows that while one could enlarge the on-board chemical system to perform both the perigee and apogee functions, the addition of an SEP system and the patience of a month to three months insertion time will allow an RLV to launch 50 to over 100 percent more net mass to GEO. The longer the insertion the smaller the on-board chemical system. Once on orbit the SEP system is now available for NSSK use as well as quick, multiple repositioning; a capability potentially valuable for both DOD and commercial users.
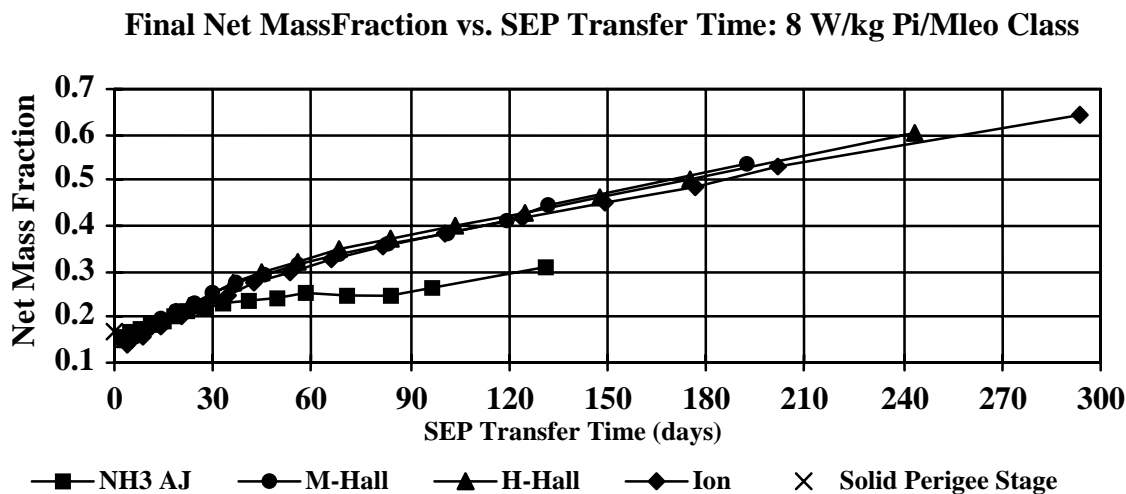
Final Net MassFraction vs. SEP Transfer Time: 8 W/kg Pi/Mleo Class

Figure 9.—Final net mass fraction vs. SEP transfer time.

CONCLUSIONS

The elimination of upper stages using on-board chemical and electric propulsion systems was examined for GEO spacecraft. Launch vehicle step-down from an Atlas IIAR to a Delta 7920 (no upper stage) was achieved using expanded on-board chemical tanks, 40 kW payload power for electric propulsion, and a 60 day elliptical to GEO SEP orbit insertion. Optimal combined chemical and electric trajectories were found using SEPSPOT. While Hall and ion thrusters provided launch vehicle step-down and even more payload for longer insertion times, NH3 arcjets had insufficient performance to allow launch vehicle step-down. Degradation levels were only 5 to 7 percent for launch step-down cases using advanced solar arrays. Results were parameterized to allow comparisons for future reusable launch vehicles. Results showed that for an 8 W/kg initial power / launch mass power density spacecraft, 50 to 100 percent more payload can be launched using this method.

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Solar Electric Propulsion (SEP) when used for station keeping and final orbit insertion has been shown to increase a geostationary satellite's payload when launched by existing expendable launch vehicles. In the case of reusable launch vehicles or expendable launch vehicles where an upper stage is an expensive option, this methodology can be modified by using the existing on-board apogee chemical system to perform a perigee burn and then letting the electric propulsion system complete the transfer to geostationary orbit. The elimination of upper stages using on-board chemical and electric propulsion systems was thus examined for GEO spacecraft. Launch vehicle step-down from an Atlas IIAR to a Delta 7920 (no upper stage) was achieved using expanded on-board chemical tanks, 40 kW payload power for electric propulsion, and a 60 day elliptical to GEO SEP orbit insertion. Optimal combined chemical and electric trajectories were found using SEPSPOT. While Hall and ion thrusters provided launch vehicle step-down and even more payload for longer insertion times, NH3 arcjets had insufficient performance to allow launch vehicle step-down. Degradation levels were only 5% to 7% for launch step-down cases using advanced solar arrays. Results were paramaterized to allow comparisons for future reusable launch vehicles. Results showed that for an 8 W/kg initial power/launch mass power density spacecraft, 50% to 100% more payload can be launched using this method.

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